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14. ABSTRACT
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University Nanosat System Thermal Design, Analysis, and Testing

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ABSTRACT

Thermal design for space systems is an iterative process that balances the temperature requirements for all mission phases with the available resources. Secondary payloads often have to be designed for a wide range of conditions available on various launch platforms, without the benefit of additional resources such as power or thermal shielding. This paper will discuss the thermal design, analysis, and thermal vacuum testing of a small satellite payload that was initially intended for launch from the US Space Shuttle and eventually launched on the EELV Heavy demonstration in December 2004.

Keywords: Satellite thermal design, thermal analysis, thermal model validation

1. INTRODUCTION

The capability provided by performing missions on earth orbiting platforms is unparalleled. Many services that are taken for granted today are enabled by satellites and their various payloads, from weather tracking and forecasting, to global telecommunications coverage, to GPS services, imaging for mapping travel routes or tracking damages caused by natural disasters, as well as basic research opportunities not possible on earth. With all the benefits and opportunities of space based operations one may wonder why more is not done in space. The simple answer is that it is difficult and expensive to get to and operate in space.

The challenges of getting into space will not be addressed in this paper, that subject will be left to the numerous publications and conferences on this subject. Suffice it to say that the technological challenges of harnessing and releasing in a controlled and useful manner the enormous amount of energy required to take mass from the surface of the earth and putting it into orbit make space launch both difficult and expensive. The scarcity of the launch resources means that that there is need to make maximum use of every opportunity that arises. For programs that would never have sufficient funding to procure a launch of their own, piggybacking as a secondary payload on the launch of something else is the only way to ever get into orbit.

Secondary payload opportunities have some unique challenges and restrictions. First and foremost whatever happens, the secondary payload cannot affect the launch and operation of the primary mission. The secondary payload is simply along for the ride. This means that when, where, and how a secondary payload is released from a launch vehicle is dictated by the primary payload. Second, launch opportunities of this type are going to happen on relatively short schedules. A mission designer is going to purchase the smallest possible launch vehicle to achieve the performance requirements, because the larger the launch vehicle the more expensive it is. That being said the mission designer holds some mass and volume margin on the launch vehicle throughout the design and manufacture of the spacecraft again because of the enormous cost of increasing the launch vehicle size and the risk of the manufactured satellite being heavier and/or larger than planned. Once the satellite is in manufacturing any remaining mass margin becomes available and will be filled by ballast or secondary payloads. Any time that margin becomes available early in the design cycle it is used either to increase payload performance or decrease the launch vehicle requirements, not to provide launch services for other missions. Finally, when opportunities arise documentation must be provided, analysis is completed, and integration takes place on the schedule dictated by the primary payload. This means that one must have a manufactured, documented, tested, easily integrated, and self sufficient payload ready for integration in a timeframe that is less than a year, and possibly only a few months.

Now that the mission can get into space, what is so difficult and expensive about operating there? The primary elements contributing to the situation include the need to operate remotely and largely autonomously, vacuum, microgravity, being subjected to elevated energy levels across the entire electromagnetic spectrum, and the thermal environment dictated by orbital dynamics. In short, the environment in space is very different than what is encountered on the earth. Designing missions and hardware that can operate in this harsh environment is in itself difficult and expensive.

This paper focuses on the thermal design, analysis, and validation of a small (less than 100kg) satellite for low earth (LEO) operations. The approach and results will be discussed, along with the contributing factors for the design decisions to be made. Finally, some general rules of thumb for this class of mission will be presented for use by designers.

2. UN MODELS AND RESULTS

2.1 The University Nanosat Program

To illustrate the application of the process outlined a summary of the process used for the University Nanosat (UN) first and second missions will be given. The UN program is sponsored by many government and industrial partners and is intended to provide opportunities to design, build, and launch satellites by university students and to provide flight qualification opportunities for a myriad of technologies. The universities designed and built their own small satellites and AFRL was be responsible for designing a carrier, the multi satellite deployment system or MSDS, to launch the satellites and eventually release them into their appropriate orbit.

The launch vehicle chosen for the mission was the US space shuttle and the payload carrier was the Shuttle Hitch Hiker Experimental Launch System (SHELS) adapter. While the decision resulted in a significant number of design restrictions on the system as a whole, only those that are relevant to the thermal design will be discussed. This choice provided limitations on the orbit that could be achieved with no additional boost stage, weight and size limitations, and numerous safety restrictions.

The mission design focused on in this effort was primarily on phases when the satellites are not operational. The mission parameters were intentionally chosen such that prior to release from the MSDS all the satellites would be inhibited from any operation by power system safeties, and once released the MSDS with mounted satellites must remain together and operationally inhibited to the maximum extent possible for a period of 2-3 days to ensure that the shuttle final separation cannot result in a collision with the shuttle. These choices minimized the safety requirements placed on the satellites for shuttle manifesting, reducing the design, testing, and documentation requirements while simultaneously maximizing the launch opportunities. AFRL's thermal modeling was done using the Sinda/Fluint software.

2.2 Common Component Details

The MSDS was designed first for mechanical properties and secondly for thermal properties. The thermal model inputs for the MSDS are shown in Table 1. The MSDS was a machined aluminum plate that mounted to the SHELS adapter, and to the satellites via the low shock stack separation system (SSS). The MSDS also contained a battery and timing circuits that would remove inhibits and release the satellites after a set period of time from the release from the shuttle. The MSDS had to be designed such that it would enable the appropriate operation of the MSDS and not exceed the temperature limits of these systems.

Table 1. MSDS Thermal Model Properties

MSDS	Mass (kg)	Thickness (m)	Width or Height (m)	Length or Diameter (m)	Radius (m)	Node #'s	Material	Density (kg/m3)	Conductivity (W/mK)	Specific Heat (J/kgK)	Coating	Abs	Emis	Conductor (W/K)	Comments
Top Plate		0.0095	0.9144	0.508		19-21	Aluminum	2800	167.9	962.9	Anodized Black	0.88	0.88		
Bottom Plate		0.0095	0.4445	0.56515		16-18	Aluminum	2800	167.9	962.9	Anodized Black	0.88	0.88		
Ribs		0.019	0.889	0.0889		10-15	Aluminum	2800	167.9	962.9	Anodized Black	0.88	0.88		
Ring		0.0254	0.0889	0.0889	0.044	3-9	Aluminum	2800	167.9	962.9	Anodized Black	0.88	0.88		
Battery	4.536					30-37	Aluminum	2800	167.9	962.9	Anodized Gold	0.84	0.35	0.00147	Battery to MSDS bottom conductor based on 1cm G-10
SHELS		0.00762	0.0762	0.4064	0.203	1-2	Aluminum	2800	167.9	962.9	Anodized Gold	0.84	0.35		

The stack separation system (SSS) was the interface between the satellite, or satellites, and the MSDS. Its primary components are a set of springs to provide separation force, and a mechanism that retained a clamped band with a Kevlar loop that would be cut at the appropriate time by melting through it. From a thermal standpoint the design had to be robust enough to not ever release prior to the appropriate time, either by getting too hot or too cold, and then to have sufficient power available to cut the Kevlar in the coldest possible configuration. At the same time it was highly desirable to minimize the power required to actuate the SSS so that the battery capacity (and mass) could be minimized.

The second type of separation system is the LiteBand. It was also provided by AFRL as a low shock separation system but it is only used between satellites in a stack. Packaging requirements make the design and actuation scheme somewhat different, and from a thermal view it is not as critical. It was analyzed primarily as a thermal conductor between satellites and as long as the satellites remained within their appropriate temperature ranges there were no thermal concerns.

The SHELS adapter was generally ignored in most of the analysis. Conduction to it and the shuttle were considered negligible while in the shuttle bay. Radiation to the thermal bay and to the environment during the mission phase in the shuttle was the primary concern. Being a secondary payload looking for the most launch opportunities forced us to consider a wide range of potential orientations. Fortunately the shuttle itself cannot operate for more than 30 minutes out of every couple hours payload bay facing the sun, nor is it generally desirable to have the payload bay face deep space for more than 1 or 2 hours at a time. This is because the shuttle bay doors are the thermal radiator for the orbiter and it also has systems that must maintain temperatures similar to that of the UN payloads. Cold shuttle orientations require extra heater power to keep systems warm, and sun views limit the number of systems that can be operated, both limiting mission utility so solutions other than facing the bay to either f these extremes are generally sought first.

A thermal shroud was to be used in the shuttle bay to protect the UN payload from thermal loads induced by other payloads. The shroud design was the responsibility of NASA engineers and inputs were provided to make this possible. The shroud would go around the sides of the UN envelope and be mounted to the SHELS. The “top” of the shroud would be open (no cover) to allow for release of the UN payload, and the “bottom” would also be open, with the MSDS and SHELS adapter as the thermal radiation barriers. A secondary benefit of the thermal shroud was that it would provide a touch barrier for the astronauts, protecting them from hot or sharp edges and surfaces, and protecting the satellites surfaces from damage if touched or grabbed accidentally.

2.3 UN1 Details

The UN1 mission consisted of three satellites one cube shaped from called Orion, and a set of two sextagonal satellites called Emerald, with the individual satellites called Chromium and Beryl. The satellites were designed and built by Stanford and Santa Clara University. The thermal model inputs used in the analysis are shown in Table 2. Note that the thermal capacity is based the assumption that all mass is aluminum. Surface absorptivity and emissivity was calculated based on a weighted average of the surface coatings, when applicable.

Sensitive components were assigned nodes with appropriate properties, and their temperature limits are shown in Table 3. Temperature limits are based on three different modes. Operating temperature refers to what is required for the component to function properly. Non-operating temperature refers to what temperatures are allowable for a limited amount of time when the system or component is not operational, but when temperatures are returned to the operational range they are expected to once again be able to operate normally. Storage temperatures are meant to reflect the maximum temperature that a system or component can be exposed to without catastrophic failures; however, if the temperatures do exceed the non-operational range and go into the storage range components will have to be re-furbished or replaced to reclaim operability.

The UN1 thermal model is shown in figure 1. Note that the Emerald satellites have been further simplified from the sextigon to a cylinder with 6 nodes and circular top and bottom plates. This simplified the modeling of conduction between the faces and minimized the number of nodes and conductors that would have been necessary if the actual shape was used. Some simple analyses showed that the geometric representations were equal in the model.

Table 2. UN1 Thermal Model Inputs.

	Mass (kg)	Height (m)	Width or Diameter (m)	Length or Radius (m)	Panel Thickness (m)	Panel Weight (kg) based on Al	Lumped Node Weight (kg)	Thermal Mass (J/K) based on aluminum	Node #'s	Material	Density (kg/m3)	Conductivity (W/mK)	Specific Heat (J/kgK)	Coating	Abs	Emis
Orion	35	0.4445	0.4445	0.4445	0.00127	4.2	30.8	29642.3	100-106 Lumped Mass 106	Aluminum	2800	167.9	962.9	Solar Cells + Black Paint	0.767	0.83
Orion SSS	1.438	0.1072	0.400558	0.200279	0.00381				46-51	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
Beryl	15.44	0.8128	0.468122	0.234061	0.00127	5.5	4.3	4105.9	120-128 Lumped Mass 128	Aluminum	2800	167.9	962.9	Solar Cells	0.82	0.87
Beryl Battery		0.0889	0.1524	0.0762	0.00635		2.7	2599.8								
Beryl Colloid Thruster							0.2	192.6								
Beryl VLF	0.1						0.1	96.3								
Beryl Liteband	1	0.054	0.395948	0.197974	0.00254				60-65	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
Chromium	15.44	0.8128	0.468122	0.234061	0.00127	5.5	4.3	4105.9	110-118 Lumped Mass 118	Aluminum	2800	167.9	962.9	Solar Cells	0.82	0.87
Chromium Battery		0.0889	0.1524	0.0762	0.00635		2.7	2599.8								
Chromium Colloid Thruster							0.2	192.6								
Chromium VLF	0.1						0.1	96.3								
Chromium SSS	1.438	0.1072	0.400558	0.200279	0.00381				40-45	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03

Table 3. UN1 Temperature Limits

	Min Operating (C)	Max Operating (C)	Min Non-Operating (C)	Max Non Operating (C)	Min Storage (C)	Max Storage (C)
Emerald Batteries	0	50	-30	50	-30	50
Emerald Colliod Thrusters	0	50	-20	50	-20	50
Emerald VLF's	-25	85	-25	85	-25	85
Orion Battery	0	50	-30	50	-30	50
Orion Prop Valves	-40	100	-40	80	-40	100
MSDS Battery	-20	30	-30	40	-30	60
SSS CBOD	-40	70	-40	70	-40	70

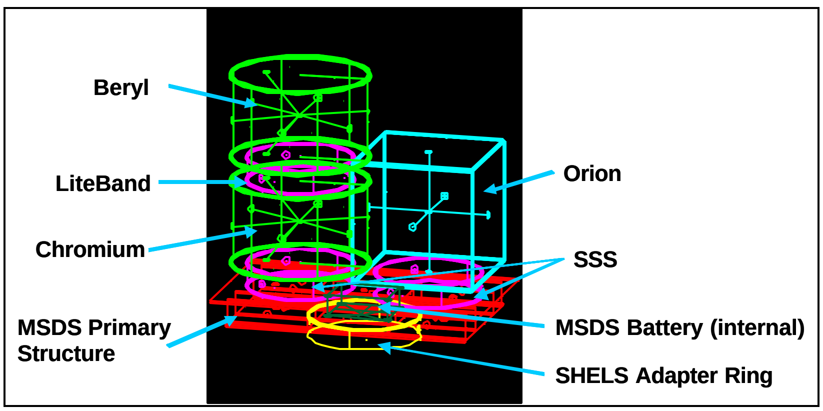

Figure 1. UN1 Graphical Thermal Model

2.4 UN2 details

The UN2 mission consisted of two sacks of three satellites each. One stack, named Ionospheric Observation Nanosat Formation (ION-F), consisted of satellites from Utah State University, the University of Washington, and Virginia Polytechnic University. The second stack, named three corner sat (3CS), consisted of satellites from New Mexico State University, the University of Colorado, and Arizona State University. All satellites were sextagonal shaped. Figure 2 shows the thermal model for UN2. The replacement of the sextigons with cylinders was used again. The thermal model properties are shown in Table 4, and the temperature limits are shown in Table 5.

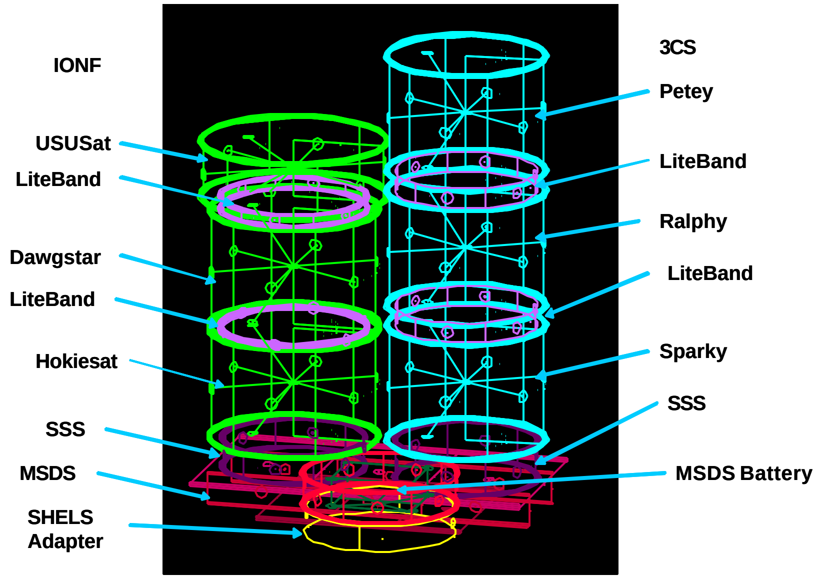

Figure 2. UN2 Graphical Thermal Model

2.5 Analysis Parameters

The thermal analyses were conducted for range of orbits and orientations. The shuttle generally operates around a beta angle of 30°, however it can vary around that depending on the mission specifics and the season. The highest expected beta angle was for a space station mission at nearly 60°. For the analyses beta angles of 0°, 30°, and 60° were used. As for the orientation after release from the shuttle it was not expected that a strictly inertial orientation could be achieved, but that instead the system would tend to tumble and could eventually spin about a single axis after a time period much longer than the satellites would remain mounted to the MSDS. A tumbling system would have a more thermally equal and stable temperature; however on the off chance that the stack deployed into an inertial orientation for the short duration it was decided that temperature limits would be based on analysis for an inertial state with either the MSDS facing the earth (referred to as the nadir orientation), or the satellite stacks would be inertially facing the earth (the anti-nadir orientation). For the approximately 280km altitude orbit and the beta angles this would represent the temperature extremes possible.

Generally analyses started with all nodes at 20C and a transient orbital analysis was run for a 24 hour period, or however long was required for the temperature cycles between orbits to stabilize. A summary of temperatures was then reported based on the minimum or maximum experienced during a steady state cycle.

Table 4. UN2 Thermal Model Inputs.

IONF	Mass (kg)	Height (m)	Diameter (m)	Radius (m)	Panel Thickness (m)	Panel Weight (kg) based on Al	Lumped Node Weight (kg)	Thermal Mass (J/K) based on aluminum	Node #'s	Material	Density (kg/m3)	Conductivity (W/mK)	Specific Heat (J/kgK)	Coating	Abs	Emis
USUSat	13.84	0.152	0.50165	0.25083	0.00254	4.52	9.32	8977.88	150-158 Lumped Mass 158	Aluminum	2800	167.9	962.9	Misc GaAS, White Paint, Alum, Black Delrin	Side 1:0.75 Side 2:0.23 Side 3:0.569 Side 4:0.569 Side 5:0.569 Side 6:0.23 Nadir:0.497 Zenith:0.555	Side 1:0.87 Side 2:0.88 Side 3:0.578 Side 4:0.578 Side 5:0.578 Side 6:0.88 Nadir:0.386 Zenith:0.534
Liteband	1	0.035	0.395948	0.19797	0.00254				86-91	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
UW (Dawgstar)	15	0.322	0.4572	0.2286	0.00254	5.62	9.38	9034.70	140-148 Lumped Mass 148	Aluminum	2800	167.9	962.9	Misc GaAS, White Paint, Alum	Side 1:0.614 Side 2:0.614 Side 3:0.614 Side 4:0.614 Side 5:0.614 Side 6:0.614 Nadir:0.555 Zenith:0.467	Side 1:0.682 Side 2:0.682 Side 3:0.682 Side 4:0.682 Side 5:0.682 Side 6:0.682 Nadir:0.534 Zenith:0.312
Liteband	1	0.01	0.395948	0.19797	0.00254				80-85	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
VPI (Hokiesat)	17.01	0.298	0.4572	0.2286	0.00254	5.38	11.63	11200.26	130-138 Lumped Mass 138	Aluminum	2800	167.9	962.9	Misc GaAS, White Paint, Alum	Side 1:0.614 Side 2:0.614 Side 3:0.614 Side 4:0.614 Side 5:0.614 Side 6:0.614 Nadir:0.230 Zenith:0.379	Side 1:0.682 Side 2:0.682 Side 3:0.682 Side 4:0.682 Side 5:0.682 Side 6:0.682 Nadir:0.88 Zenith:0.09
SSS		0.082	0.400558	0.20028	0.00381				46-51	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
3CS	Mass (kg)	Height (m)	Diameter (m)	Radius (m)	Panel Thickness (m)	Panel Weight (kg) based on Al	Lumped Node Weight (kg)	Thermal Mass (J/K) based on aluminum	Node #'s	Material	Density (kg/m3)	Conductivity (W/mK)	Specific Heat (J/kgK)	Coating	Abs	Emis
NMSU (Petey)	16	0.318	0.429336	0.21467	0.00254	5.10	10.90	10493.38	120-128 Lumped Mass 128	Aluminum	2800	167.9	962.9	Top:GaAs+Black Paint Sides:GaAs Bottom:Bare Al	Top:0.942 Sides:0.92 Bottom:0.16	Top:0.872 Sides:0.85 Bottom:0.03
Liteband	1	0.054	0.395948	0.19797	0.00254				66-71	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
CU (Ralphie)	16	0.318	0.429336	0.21467	0.00254	5.10	10.90	10493.38	110-118 Lumped Mass 118	Aluminum	2800	167.9	962.9	Top:GaAs+Black Paint Sides:GaAs Bottom:Bare Al	Top:0.942 Sides:0.92 Bottom:0.16	Top:0.872 Sides:0.85 Bottom:0.03
Liteband	1	0.054	0.395948	0.19797	0.00254				60-65	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03
ASU (Sparky)	16	0.318	0.429336	0.21467	0.00254	5.10	10.90	10493.38	100-108 Lumped Mass 108	Aluminum	2800	167.9	962.9	Top:GaAs+Black Paint Sides:GaAs Bottom:Bare Al	Top:0.942 Sides:0.92 Bottom:0.16	Top:0.872 Sides:0.85 Bottom:0.03
SSS		0.107	0.400558	0.20028	0.00381				40-45	Aluminum	2800	167.9	962.9	Bare Al	0.16	0.03

Table 5. UN2 Temperature Limits.

	Min Operating (C)	Max Operating (C)	Min Non-Operating (C)	Max Non Operating (C)	Min Storage (C)	Max Storage (C)
ION-F: USU Sat (Battery)	0	30	-20	30	-20	30
ION-F: Dawgstar	-20	50	0	70	-20	30
ION-F: HokieSat	10	60	0	70	-20	30
3CS:	-33	57	-33	57	-33	57
MSDS Battery	-20	30	-30	40	-30	40
SSS CBOD	-40	70	-40	70	-40	70

2.6 Shuttle Launch Results Summary

The first analysis trade done for the UN1 mission was to determine the appropriate coating for the MSDS plate. Similarly, trades were done on the MSDS battery box for how it would be thermally mounted to the MSDS. A robust and easily manufactured coating was desired for the MSDS exterior surfaces. The MSDS plate was to be handled often and could be exposed to repeated mounting, removal, and remounting of satellite stacks as integration and testing was conducted.

First analysis showed that the thermal properties of bare aluminum, or aluminum with a clear anodize finish would be unacceptable hot. Next trades were done to quantify the effects of a white paint versus a black paint finish. Results are shown in Tables 6 and 7, with typical temperature plots are shown in Figures 3 and 4. While black paint has a slightly larger orbital temperature variation, it is not nearly as cold as white paint, so a black coating was chosen. Additional analysis on various black coatings showed that a black anodize could be used with essentially the same results, so a black anodized coating was chosen for the MSDS. The solution with a black paint remains warmer because of the absorption of thermal energy from the sun and earth.

The battery box inside the MSDS was completely enclosed inside the MSDS. A design that further insulated the battery from the MSDS structure was chosen so that the battery would remain as warm as possible. A box made with G-10 and G-10 spacers was chosen with a gold kapton surface finish that represented the kapton tape that the box would be covered in for both thermal and electrical reasons. The G-10 minimizes thermal conduction to the battery box, and the gold kapton minimizes thermal emission from the battery to the MSDS internal surfaces while increasing absorption of thermal energy from the MSDS when it is sufficiently warm.

MSDS with Black Paint

	Nadir Max (C)	Nadir Min (C)	Anti-Nadir Max (C)	Anti Nadir Min (C)
MSDS	22	-38	40	-34
MSDS Battery	-4	-4	-2	-2
Top Emerald	5	-12	0	14
Bottom Emerald	0	-12	0	-14
Orion	-2	-18	-2	-18

Table 6. Temperatures With Black MSDS.

MSDS with White Paint

	Nadir Max (C)	Nadir Min (C)	Anti-Nadir Max (C)	Anti Nadir Min (C)
MSDS	-7	-35	-14	-44
MSDS Battery	-13	-13	-22	-22
Top Emerald	17	-2	-12	-22
Bottom Emerald	-9	-20	-17	-27
Orion	-13	-26	-21	-32

Table 7. Temperatures With White MSDS.

For UN1 a detailed analysis of the SSS release mechanism was conducted to ensure that it would not accidentally release too soon, and to determine the power required to actuate at the appropriate mission time from the worst case cold temperature. A sensitivity analysis was used to determine the maximum temperature of the release mechanism. It was shown that even if this unit had worse than expected thermal contact through its bolted joint to the MSDS, all it's metal surfaces were bare anodized aluminum, and it was stuck in an orientation where it was solar inertial (faces the sun during illumination and the earth during eclipse) it was impossible for the mechanism to ever have a thermal gradient exceeding 4C from the temperature of the MSDS plate. The maximum temperature in this worst case configuration is more than 30C from the actuation temperature of the mechanism so it would be thermally impossible to actuate the SSS. Additionally, on the cold side the mechanism cannot vary much from the temperature of the MSDS plate so the power system design assuming that it would actuate at a temperature of -40C was sufficient to ensure success.

Finally, the minimum and maximum temperatures for the UN1 system were analyzed for the above orientations and orbits. The results are presented in Table 8 and show that post shuttle ejection the satellites remain within their non-operating temperature rages. Operational analysis of the satellite thermal designs were conducted by the universities, and pre-shuttle deployment analyses were done by NASA. Results from these parties indicated that the thermal designs were sufficient for safe launch and operation.

For UN2 similar analyses and results were generated. These are presented in Table 9.

In May 2002 thermal balance testing was conducted on the UN2 assembly. The results validated the thermal models and designs. It is expensive, and often impossible to simulate orbital conditions in thermal vacuum because of the varying and diverse sink temperatures of the sun, earth, and deep space. Efforts were not made to match the orbital environment exactly, but instead the test conditions were analyzed, and the responses were as expected.

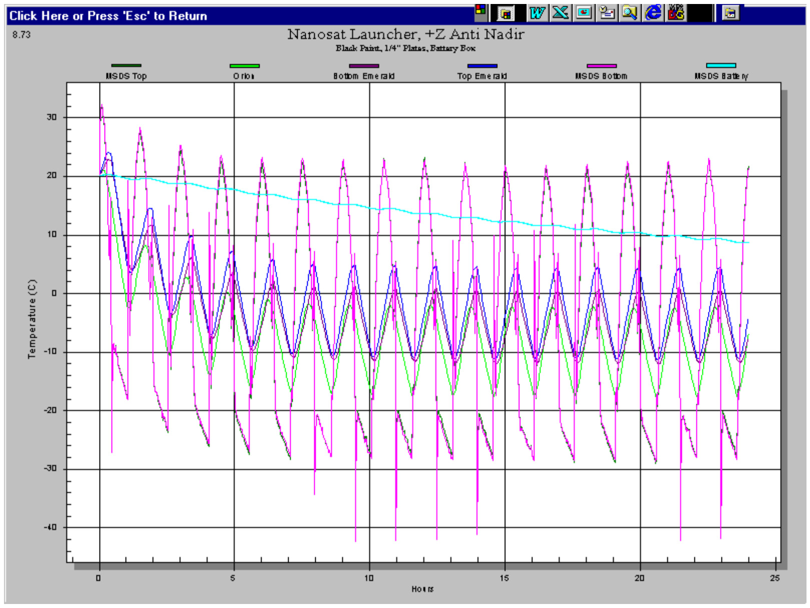

Figure 3. Transient Model Temperature Summary, MSDS With Black Paint.

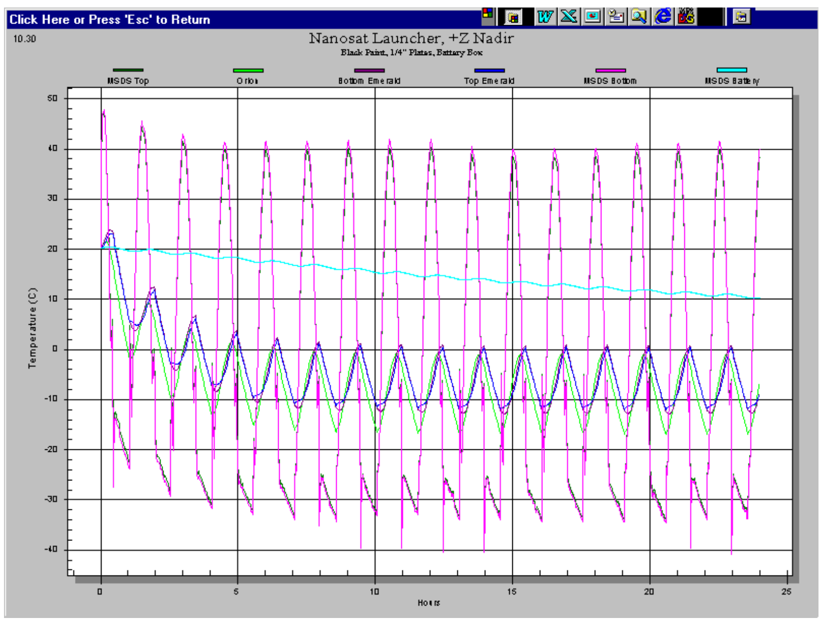

Figure 4. Transient Model Temperature Summary, MSDS With White Paint.

Table 8. UN1 Temperature Limit Summary.

	Beta 0				Beta 30				Beta 60			
	Nadir		Anti-Nadir		Nadir		Anti-Nadir		Nadir		Anti-Nadir	
	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max
MSDS	-13	-1	-13	-7	-4	7	-2	5	1	9	6	12
MSDS Battery	-7	-6	-10	-9	1	2	1	2	5	6	8	9
Orion	-16	-11	-17	-10	-2	5	-3	6	6	12	5	13
Chromium	-17	-1	-17	-1	-6	10	-7	11	4	20	3	20
Beryl	-17	-1	-20	6	-8	10	-13	13	2	20	-4	17

Table 9. UN2 Temperature Limit Summary.

	Beta 0				Beta 30				Beta 60			
	Nadir		Anti-Nadir		Nadir		Anti-Nadir		Nadir		Anti-Nadir	
	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max
MSDS	-10	5	-12	-4	-2	12	0	9	7	17	13	20
MSDS Battery	-3	-2	-8	-7	5	6	4	5	12	13	15	16
3CS Sparkie	-14	-6	-15	-6	0	10	0	11	14	26	15	27
3CS Ralphie	-15	-6	-13	-4	0	11	-1	12	15	28	14	27
3CS Petey	-14	-3	-12	8	0	13	-3	17	13	26	8	23
IONF Hokiesat	-11	-1	-13	-6	0	9	0	7	10	17	12	20
IONF Dawgstar	-15	-7	-16	-7	-3	7	-3	7	9	18	9	19
IONF USUSat	-14	-7	-17	-4	-3	5	-4	7	8	15	6	15

2.7 UN2 Modified Heavy Launch Demo

Launch opportunities other than the shuttle have always been a window of opportunity for the UN program, and after the Columbia disaster in 2002 it became the primary focus of the AFRL team. With the aid of the US Air Force Space Test Program (STP) an opportunity was found in late 2002 aboard the Delta 4 first flight, also referred to the EELV Heavy Launch Demonstration. The demo was to show GEO orbit launch and insertion, but the UN payload could be attached and released in LEO before the upper stage was fired. Only the 3CS team decided to participate in the opportunity so the system was reconfigured to have one satellite with a mass shim on one SSS, and the other SSS would have a stack of only two satellites. Thermal analysis was re-done to ensure that the system still worked, including making sure that the SSS mechanism would not release inadvertently in the new configuration. Figure 5 shows the UN2 modified thermal model. Note that while the figure shows a very simple model that was used by the launch provider for their system level analysis, models were generated and analyzed to confirm that this simple model was representative. The post deployment temperature summary is shown in Table 10.

While the UN2 modified mission launch failed to achieve the predicted orbit insertion, and thus had no time to operated before it re-entered the earths atmosphere approximately 30 minutes after deployment, based on the analysis and limited testing we expected a fully functional mission.

3. CONCLUSION

The thermal design and analysis for the UN mission was successfully completed. A robust design of the launch platform was generated, tested, and showed operability on two different launch vehicles. The thermal design kept the satellites and deployment platform within required temperature ranges throughout all mission phases, from launch and initial deployment, to the three day waiting period between initial deployment and satellite release, to operation of the satellites on orbit.

The design effort was iterative and required active participation by many team members. Understanding the design space, mission requirements and limitations, and effective communication were keys to the success. Communicating how design and component choices affected the thermal design and response was also essential to making the most efficient system design choices. In some cases components were chosen with wider temperature ranges, reducing risk and power requirements; and in other situations cost and handling capability was chosen instead of an enhanced thermal design.

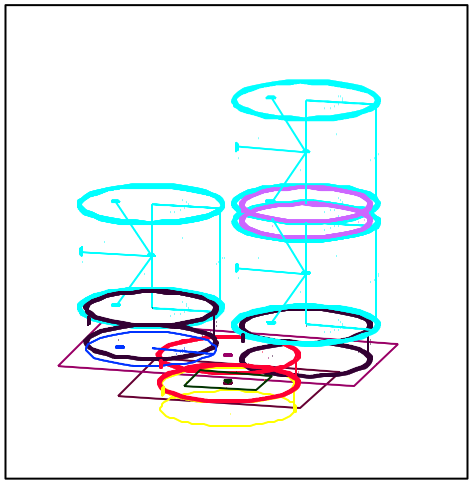

Figure 5. UN2 Modified Thermal Model For The Heavy Launch Opportunity.

Table 10. UN Modified Temperature Summary.

	Beta 0				Beta 30				Beta 60			
	Nadir		Anti-Nadir		Nadir		Anti-Nadir		Nadir		Anti-Nadir	
	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max	Min	Max
MSDS Battery	4	5	0	1	8	9	6	7	10	11	15	16
3CS Sparkie	-6	5	-10	7	3	15	0	17	11	24	10	24
3CS Ralphie	-6	2	-7	3	2	12	1	12	11	24	11	24
3CS Petey	-8	2	-7	13	0	15	-2	17	9	23	6	21

This success has resulted in the generation of some general guidance regarding UN mission in the future, and could have general applicability for similar configurations. These are included here only as general rules of thumb, but caution must also be used because for every rule it is possible to come up with situations where it will not work.

- Small satellites in LEO are generally power limited. To minimize heater power requirements black coatings, or those with similar high absoprivity and high emissivity values, are desirable.
- Whenever possible choose components that have operational temperature ranges of -10C to +60C, and non operational temperatures of -40C to +100C. This covers what can generally be expected for operational phases, and provides a good safety margin for non-operational phases.
- Active deployment mechanisms should have an operational (non-deployment) temperature range of at least -50C to +150C. This is for the protection of the launch vehicle and primary payload.
- MLI can be used to limit radiation internally and externally. Beta cloth can also be used as long as the mission is less than a year; however it is not a readily available as MLI.
- Pay attention to conduction, both in component designs and across interfaces. Aluminum is a great material for conduction, due to its high thermal conductivity to mass ratio as well as it's excellent strength and manufacturability. Limiting conduction paths through interface areas and materials like G-10 can help if low conductivity is needed.
- One should not need heat pipes, capillary pumped loops, pumped fluid loops, peltier coolers, louvers, heat pumps, or cryocoolers. Either there is not sufficient power available in the satellite to operate these devices, or there is not enough power available to a payload to generate sufficient waste heat to require them.
- Packaging components together will allow a designer to take advantage of the thermal mass of the whole system and minimize heater power. When this is insufficient thermal mass can be added with phase change materials like paraffin's. Caution must be used with phase change materials though, they generally have poor thermal conductivity so getting heat into and out of them could be a limiting factor.

References:

Additional information on the UN satellites can be obtained from the participating universities.

Information on the design requirements for the shuttle can be obtained from NASA.

Information on the UN2 Heavy Launch Demonstration can be found at
http://telemetry.nmsu.edu/new_page.html and
http://www.spaceflightnow.com/delta/d310/041201demosat.html.

Sinda/Fluint software information is available from Cullimore and Ring Technologies, www.crtech.com.